NASA/TM—1998-206222

RHETT/EPDM Performance Characterization

T. Haag
Lewis Research Center, Cleveland, Ohio

M. Osborn
Naval Research Laboratory, Washington, D.C.

June 1998

The NASA STI Program Office . . . in Profile

Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA Scientific and Technical Information (STI) Program Office plays a key part in helping NASA maintain this important role.

The NASA STI Program Office is operated by Langley Research Center, the Lead Center for NASA's scientific and technical information. The NASA STI Program Office provides access to the NASA STI Database, the largest collection of aeronautical and space science STI in the world. The Program Office is also NASA's institutional mechanism for disseminating the results of its research and development activities. These results are published by NASA in the NASA STI Report Series, which includes the following report types:

- TECHNICAL PUBLICATION. Reports of completed research or a major significant phase of research that present the results of NASA programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA's counterpart of peer-reviewed formal professional papers but has less stringent limitations on manuscript length and extent of graphic presentations.

- TECHNICAL MEMORANDUM. Scientific and technical findings that are preliminary or of specialized interest, e.g., quick release reports, working papers, and bibliographies that contain minimal annotation. Does not contain extensive analysis.

- CONTRACTOR REPORT. Scientific and technical findings by NASA-sponsored contractors and grantees.

- CONFERENCE PUBLICATION. Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or cosponsored by NASA.

- SPECIAL PUBLICATION. Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.

- TECHNICAL TRANSLATION. English-language translations of foreign scientific and technical material pertinent to NASA's mission.

Specialized services that complement the STI Program Office's diverse offerings include creating custom thesauri, building customized data bases, organizing and publishing research results . . . even providing videos.

For more information about the NASA STI Program Office, see the following:

- Access the NASA STI Program Home Page at ***http://www.sti.nasa.gov***

- E-mail your question via the Internet to help@sti.nasa.gov

- Fax your question to the NASA Access Help Desk at (301) 621-0134

- Telephone the NASA Access Help Desk at (301) 621-0390

- Write to:
  NASA Access Help Desk
  NASA Center for AeroSpace Information
  800 Elkridge Landing Road
  Linthicum Heights, MD 21090-2934

NASA / TM—1998-206222

RHETT / EPDM Performance Characterization

T. Haag
Lewis Research Center, Cleveland, Ohio

M. Osborn
Naval Research Laboratory, Washington, D.C.

Prepared for the
25th International Electric Propulsion Conference
sponsored by the Electric Rocket Propulsion Society
Cleveland, Ohio, August 24–28, 1997

National Aeronautics and
Space Administration

Lewis Research Center

June 1998

Available from

NASA Center for Aerospace Information
800 Elkridge Landing Road
Linthicum Heights, MD 21090-2934
Price Code: A03

National Technical Information Service
5287 Port Royal Road
Springfield, VA 22100
Price Code: A03

RHETT/EPDM Performance Characterization

T. Haag
NASA Lewis Research Center

M. Osborn
Naval Research Laboratory

Abstract

The 0.6 kW Electric Propulsion Demonstration Module flight thruster system was tested in a large vacuum facility for performance measurements and functional checkout. The thruster was operated at a xenon flow rate of 3.01 mg/s, which was supplied through a self-contained propellant system. All power was provided through a flight-packaged power processing unit, which was mounted in vacuum on a cold plate. The thruster was cycled through 34 individual startup and shutdown sequences. Operating periods ranged from 3 to 3600 seconds. The system responded promptly to each command sequence and there were no involuntary shutdowns. Direct thrust measurements indicated that steady state thrust was temperature sensitive, and varied from a high of 41.7 mN at 16°C, to a low of 34.8 mN at 110°C. Short duration thruster firings showed rapid response and good repeatability.

Introduction

The Russian Hall Electric Thruster Technology (RHETT) program is sponsored by the Ballistic Missile Defense Organization (BMDO) for the purpose of evaluating recently available Russian Hall thruster technology. Much of this hardware was developed over the past 30 years in the former Soviet Union and has seen flight application on their spacecraft. The Russian industrial establishment is now interested in marketing their space hardware to Western buyers, including the United States. Because of promising performance, the BMDO initiated the RHETT program in order to verify performance and improve compatibility with Western spacecraft.

There are a number of different types of Russian Hall thrusters available. Hall thrusters have been scaled to a wide range of power levels, ranging from hundreds of watts, to tens of kilowatts. Practical mission applications include primary orbit raising, stationkeeping, and possibly attitude control. The thruster which is the subject of this test will be operated in space at 0.6 kW, for stationkeeping and orbit raising of a US satellite. The Russian thruster was manufactured by the TsNIIMASH organization, and designated the TAL D-55. While the SPT-100 has been in use for a number of years, the Thruster with Anode Layer (TAL) has not seen flight application to date. There have been a number of variations on this design, but the thruster used in these tests was specifically designed to survive launch vibrations as well as to perform a full length mission in a space environment. The Naval Research Laboratory (NRL) is coordinating the integration of the thruster into an Electric Propulsion Demonstration Module (EPDM) to be used on the spacecraft[1]. A contract was awarded to the Primex Aerospace Co. (PAC) for the development of the Power Processing unit (PPU). An Auxiliary Interface Unit (AIU) has been developed by NRL for the purpose of interfacing between the Hall thruster system and the spacecraft computer. The AIU will communicate directly with the spacecraft computer in order to relay commands and telemetry.

In an orbit raising and stationkeeping mission, the thruster will be expected to fire for various lengths of time, and respond rapidly to on/off commands. Predictable performance is extremely important in order to maintain precise spacecraft control. The total impulse delivered from firing for a specified period of time must be accurately known. The purposes of the tests completed in this study were to perform a functional check of flight hardware to be used in the EPDM, and establish a data base of performance from which maneuver commands will be calibrated.

Apparatus

Thrust Stand:

Performance measurements were conducted under vacuum in a 4.6 m diameter by 20 m long tank. The axis of the vacuum tank was horizontal, and hardware could be accessed by opening either end cap after the facility was vented to atmosphere. Vacuum was maintained by simultaneous use of oil diffusion pumps and helium cryogenic panels. Pumping speed was sufficient to maintain background pressure in the facility on the order of 3 $\times 10^{-4}$ Pa during thruster operation.

Thrust measurements were made with a calibrated displacement type thrust stand, supported in an inverted pendulum configuration. The thrust stand could translate horizontally up to 5 mm in response to thrust force. Eight structural flexures restricted all motion except for that along the thruster axis. Displacement was measured using a linear variable displacement transformer (LVDT) to a resolution of 2 x 10^{-3} mm. The rectified output from the LVDT was connected to a strip chart recorder, where displacement was interpreted as measured thrust. The mechanical sensitivity of the thrust stand could be adjusted by exchanging load springs of different elastic stiffnesses. Thrust stand sensitivity was accurately determined during calibrations performed in-place and under vacuum. Three 1.65 gram weights were suspended from a mono-filament thread and applied tension along the thrust axis using a precision pulley. A stepping motor was used to raise and lower the string of weights, thereby engaging each weight in succession. Deflections resulting from the weights were the basis for the calibration, and this procedure was frequently carried out between thruster firings as a standard by which force measurements were quantified.

Because the mounted thruster assembly had a thrust-to-weight ratio of about 1.3 x 10^{-3}, the test installation was very sensitive to facility disturbances. This effect resulted from a small component of the mounted weight being resolved along the sensitive axis of the thrust stand. Thermally induced distortion within a vacuum tank is a common source of angular disturbance, and often occurs in facilities using oil diffusion pumps or cryogenic panels. A gravitational inclinometer capable of resolving 1 x 10^{-3} degrees was mounted to the base of the thrust stand and was used to measure angular deviations. The deviations were corrected using a fine pitch adjustment screw acting through a lever arm. A stepping motor was used to turn the adjustment screw, and the thrust stand base could maintain a fixed orientation as best resolved by the inclinometer.

Every effort was made to minimize friction within the thrust stand. The flexures were designed to produce very repeatable results and transient disturbances would require on the order of 100 oscillations for motion to damp out by friction. Because some of the proposed thruster firings were as short as 3 seconds, it was advantageous to have critically damped thrust stand movement. This was accomplished by using an electromechanical damping system. An operational amplifier circuit took the time derivative of the displacement signal, amplified it, and drove current through a magnetic coil. This reacted against a permanent magnet attached to the thruster mount, resulting in an idealized damping force. The circuit was only active during transients, and did not affect steady state measurements.

Xenon propellant was delivered to the thruster through a 3 mm diameter nylon tube. The tube was used as a propellant flexure, which consisted of a 4 loop coil approximately 10 cm in diameter, coaxial with the thrust axis. The tube provided a flexible connection between the stationary propellant storage assembly and the movable thruster assembly.

Electrical power was transferred from the stationary facility to the mounted thruster using two laminated multi-conductor ribbon cables. Both ribbon cables contained 30 flat conductors, each equivalent to a 24 gage wire. The conductors were arranged side by side, and sandwiched between two sheets of kapton insulation. A total of sixty individual conductors were organized into 22 circuits which powered the thruster and operated the propellant feed system. Conductors were connected in parallel such that current through each was no more than 2 A. Connections were potted with silicone rubber.

The main thrust stand assembly was contained inside a water-cooled, copper enclosure. Water was circulated through 6 mm OD. tubes soldered to the copper enclosure and pumped through a temperature-controlled water bath. This was done to keep the flexures, LVDT, and inclinometer at constant temperature. An opening at the top of the enclosure allowed the thruster mount to protrude, on which the thruster assembly was attached.

<u>Hall Thruster</u>:

The EPDM thruster used in this test was developed into flight hardware based on the existing D-55 Russian design[2]. The walls of the discharge chamber were metallic, with no exposed discharge insulator commonly associated with Hall thrusters. Internal insulators were used to suspend the anode from the thruster body. Xenon propellant entered the discharge chamber at the base of the channel through numerous injector holes spaced along the perimeter.

A radial magnetic field was established at the exit of the discharge chamber using four electric coils. One coil was housed in the center body of the thruster, and three other coils were spaced at 120 degree intervals around the outside of the thruster. A circular magnet pole piece was located on the end of the center body, and an annular pole piece was attached to the outer three coils. The inner and outer magnet coils could be controlled independent of the discharge current.

A single hollow cathode was used with the thruster. The basic cathode design was originally developed at the Lewis Research Center (LeRC) for ion thruster applications, and was modified to match the flow rate and current anticipated for this thruster[3]. A swaged heater was wrapped around the outside of the cathode tube to preheat the porous tungsten insert material before ignition. A high voltage keeper was installed around the cathode orifice plate in order to facilitate ignition.

Xenon propellant for this test was stored in a high-pressure, spherical tank approximately 15 cm in diameter. The xenon entered a single-stage, mechanical regulator where it was reduced to a constant 248 kPa (36 psi). Regulated xenon flow was switched on or off by a solenoid firing valve before passing through the propellant flexure. Once on the thrust stand, xenon was divided into two flow paths for the main discharge and cathode. Flow metering and distribution was determined by a thermostatically controlled orifice block installed on the thruster mounting plate.

Electric power to the thruster was provided by a PPU manufactured by PAC. The switching power supply drew current from a 30 volt DC bus, and operated at approximately 20 kHz. The unit provided a fixed 300 volt DC output for the main discharge. Discharge current demand was mostly determined by the xenon propellant flow rate. The PPU also provided electrical current for the thruster magnets and cathode heater, as well as high voltage to ignite the cathode keeper.

An AIU was the primary means of interacting with the thruster. It operated all xenon propellant valves, provided commands for the power processor, and collected telemetry data from the thruster. The spacecraft computer will exchange digital information with the AIU, relaying command sequences to all systems for correct thruster function. The test engineer accessed the AIU through a desk top computer. Telemetry data collected included thruster discharge voltage, cathode current, thruster temperature, magnet temperature, regulated propellant pressure, floating potential, bus voltage, heater voltage, and magnet current. Because only the inner magnet voltage was monitored, it was not possible to determine magnet power. All telemetry was sampled and stored digitally at a rate of 2 Hz.

Experimental Procedure

The main objective of the tests was to measure thruster performance over a variety of operating periods, included 3, 10, 30, 60, 600, 1200, and 3600 seconds. Another objective of the tests was to determine the effect of thruster temperature on performance. The thruster was typically cycled several times at each operating period in order to verify repeatable performance.

To accomplish these goals, tests were conducted over two days. During the first day the thruster began cold, and tests were conducted from shorter to longer operating periods. Every operating period was separated by a minimum 8 minute hold, during which the thrust zero was checked and a cathode preheat cycle was performed. The thruster temperature gradually began to increase during the longer operating periods, but the majority of data was taken below 40°C. The second day began with a 3600 second operating period which heated the thruster to a temperature of 110°C. Tests were then conducted from longer to shorter duration periods, approximately in the reverse order of the first day. Again, every operating period was separated by a minimum 8 minute hold. Temperatures gradually began to decrease during the shorter periods, but the majority of data was taken above 100°C.

Propellant mass flow calculations were based on orifice temperature and pressure measurements recorded during each test. These data were compared with laboratory flow calibration curves for the feed system, and showed xenon consumption to be consistently 3.01 mg/s for the combined discharge and cathode flows.

Results and Discussion

Thruster performance is traditionally described in terms of specific impulse and thrust efficiency. Assuming steady state operation, specific impulse is calculated as thrust divided by propellant flow rate. The highest performance is usually obtained after starting transients have subsided. Because the tests carried out in the present work were sometimes very short in duration, non-ideal characteristics are a realistic concern. The cathode is preheated for 420 seconds prior to thruster ignition whether the thruster burn is for 3 seconds or 3600 seconds. Each cathode preheat cycle consumes about 38 kJ of energy, which is the same energy consumed in about 1 minute of steady-state thruster operation. A similar situation is encountered when propellant consumption is considered. The solenoid firing valve is opened 15 seconds before the thruster ignites, in order to provide time for the flow to equilibrate. A post-firing bleed down also occurs as propellant lines gradually depressurize. This escaping gas provides very little propulsive effect but is a necessary expenditure in order to ensure proper function of the thruster. As a result, 3 seconds of thruster operation consumes about 18 seconds worth of xenon.

Because calculations based on total consumption may cause confusingly low performance numbers, it was decided to calculate specific impulse and efficiency on steady-state values. Any future mission analysis using these results must independently account for xenon consumption prior to each discharge ignition. Average specific impulse was calculated using the average thrust for each operating period, and a propellant flow rate of 3.01 mg/s. Average thrust efficiency was calculated using only discharge power.

$$\eta = \frac{T^2}{2\dot{m}P}$$

where m is the propellant flow rate (3.01 mg/s), T is the average thrust, and P is the average discharge power. The above calculation does not account for power consumed by the magnets or cathode preheat. It also does not account for propellant bleed-down losses.

Thrust stand thermal and magnetic tares:

As mentioned previously, a 420 second cathode preheat cycle is required before thruster ignition. When this was done for the first thruster start, a small negative thrust indication was observed even before propellant began to flow. This is believed to be a result of resistive heating within the electrical flexure. The magnitude of the force was about -0.5 mN, or 1.3% of the full operational thrust, but occurred in the opposite direction. The cathode heater was powered by the PPU, which supplied a constant current of 8.5 A for 420 seconds prior to ignition. The heater current level was over three times the typical discharge current, and represented a worst case situation for the electrical flexure.

After most data had been taken, a series of diagnostic tests were conducted to determine thrust measurement uncertainties contributed by electrical inputs. With the facility under vacuum, the AIU commanded the cathode heater to operate for 420 seconds and then turn off. There was no discharge voltage or xenon propellant flow. The typical negative drift increased over the 420 seconds of heater operation until it stabilized at about -0.5 mN. When the heater was turned off, the thrust signal gradually returned to zero. The process appeared to be fairly linear with time, and after about 60 seconds the chart recorder trace was almost back to the original position.

The results noted above, and the consistent behavior of all thrust data suggested that a simple correction could be performed. For thruster operation 10 seconds or shorter, the effective zero thrust point was established immediately before xenon propellant is turned on. Because the thermal time constant associated with the electrical flexure was about 60 seconds, the thruster is on and off before the negative offset changes significantly. Thruster data taken during periods longer than 10 seconds were adjusted assuming a 60 second linear decay of the negative offset.

More extensive tests were performed with all equipment in place, however for convenience the facility was at atmosphere. In the first test, both magnets were powered up with all other thruster power off. Magnetic effects were of particular concern because the vacuum tank walls contained mild steel, and magnetic tares can be highly non-linear. Even at full current, there was no measurable thrust influence from the thruster magnets. In another test, the thruster discharge was electrically shorted and 2 A of current was sent through the electrical flexure. Again, there was no significant thrust effect observed. In a third test the cathode heater was electrically shorted, and 8.5 A of current were sent through the flexure. In this test a slight negative thrust drift was observed which reached a maximum of about -0.3 mN. While this tare was smaller than -0.5 mN obtained under vacuum with an actual heater, it is reasonable to conclude that resistive heating in the electrical flexure was the cause.

Performance measurements:

Thruster testing consisted of 34 individual startup and shutdown sequences. Figures 1 and 2 show the discharge voltage and current respectively over a typical 60 second operating period. As shown in the figures, voltage rises nearly instantaneously to approximately 300 volts and sustains that level for 60 seconds. The data shows about 5% scatter, however this appears to be random with no change in mean value. After the operating period expired, the voltage drops very rapidly, with some capacitive effects toward the end. Discharge current shows a similar result as the thruster is powered up. Current rapidly rises to 2.05 A and gradually declines by about 2% before the discharge is terminated. The thruster temperature during this test was 31°C.

Figure 3 shows the thrust stand chart recording during a typical 60 second operating period. The thrust stand initially is at rest, and the recording trace indicates zero. When the cathode heater is turned on 420 seconds before firing, indicated thrust gradually drifts negative to settle at approximately -0.5 mN. Xenon flow is initiated 15 seconds before firing, indicating a positive thrust of 0.2 mN. When the thruster ignites, the trace very quickly rises to peak, and then drops slightly to a value of 40.3 mN. This is followed by a gradual 4% decline to 38.7 mN before the discharge is terminated. The thrust trace goes slightly negative before rising to about 0.2 mN. Thrust approaches zero as excess xenon vents from the propellant lines. The process repeats as the cathode heater is powered in preparation for the next operating period.

In order to determine the total delivered impulse from each operating period, the thrust data in Figure 3 was corrected for zero offset and curve fit with a simple mathematical function. The function was then integrated and evaluated between the beginning and end points. 60 seconds of thrust produced a total of 2.38 N-s for the example above. Three other firings yielded impulses of 2.38, 2.37, and 2.36 N-s at temperatures of 33°C, 35°C, and 37°C, respectively.

The average thrust over 60 seconds was determined by dividing the total delivered impulse by the on-time duration, and was calculated to be 39.6 mN. This yielded an average specific impulse of 1341 s. Average thrust efficiency was determined to be 43% based solely on discharge power.

The next day the thruster was tested for an identical 60 second duration, but at temperatures of 118°C and 114°C. Voltage was consistently 300 V, but current had dropped an average of 4% to 1.91 A. Thrust and delivered impulse were also reduced by about 5% at higher temperatures, and as a consequence efficiency was 40%. The cause of lower performance at higher temperature is not fully understood. Both thrust and discharge current data would suggest that propellant flow may have been reduced at higher temperatures. Data obtained from the propellant feed system, however, does not support this hypothesis. The regulator down stream pressure was consistently maintained between 251 and 252 kPa. The thermostatically heated orifice block temperature was similarly constant. Because magnet current was also within bounds, a remaining possibility includes some type of temperature sensitive cathode or anode effect. Exploration into the kinetics of this phenomena goes beyond the scope of this work.

The shortest period of thruster operation tested was 3 seconds. Figure 4 shows a plot of discharge voltage as a function of time during such a burn, with the thruster temperature at 22°C. Because the sample rate of the data acquisition system was only 2 Hz, it would not be justified to conclude more than the average thruster voltage was 298 V. The voltage returned to zero once the discharge was terminated, with the same characteristics seen in Figure 1. Discharge current is plotted in Figure 5, and shows a similar result. The average current during 3 seconds of discharge was 2.08 A. The thrust stand response to a 3 second burn is shown in Figure 6. Because of the accelerated chart speed, the time at which xenon flow was initiated precedes the chart boundary. The thrust stand had a natural frequency of about 1 Hz, and was critically damped. The trace in Figure 6 does not stabilize until about 1.5 seconds after the thruster is turned on. This leaves only another 1.5 seconds to interpret the thrust output. The steady value is 40.7 mN, from which it can be assumed to deliver 0.122 N-s of impulse. Thrust efficiency based on discharge power was determined to be 44% .

The above measurements are compared to results of a 3 second operating period at a temperature of 89°C. Discharge voltage was an average of 300 V, but current was reduced by almost 5 % to 1.98 A. Thrust was 39.2 mN, which is a reduction of almost 4 % compared to the 22°C temperature results. Delivered impulse was proportionally less, at a value of 0.118 N-s. Because discharge power dropped as well as thrust, the efficiency was calculated to be only slightly less at 43 %.

The longest operating period was 3600 seconds (1 hour). The initial thruster temperature was 8°C, but increased steadily to 110°C toward the end of the period. Thruster voltage is shown in Figure 7 over the entire period. The voltage began at 300 V and stayed constant throughout the test except for sporadic excursions. Discharge current (Fig. 8) began at 2.05 A but dropped 7% to 1.90 A within the first 200 seconds. Current unexpectedly recovered to about 2.0 A only 5 minutes from the end of the test. This recovery coincided with experimental variations in PPU supply voltage from 30 V to 25 V, and finally 35 V. Because the thruster voltage remained constant at 300 V throughout this variation, the significance of the current drift remains unclear. The thrust trace in Figure 9 shows a similar decline early in the test, but without recovery. Thrust began at 40.7 mN at the beginning, and asymptotically approached a final value of 34.8 mN before termination. This represents a 14 % drop in thrust, which decreased efficiency at termination to only 34 %. Average thrust efficiency for the entire 3600 seconds was 38 %. Figure 9 more clearly shows the initial thrust drift of -0.5 mN resulting from cathode preheat before thruster ignition. The post test zero returned very close to the original location. The total delivered impulse after 3600 seconds was 123 N-s, and the average thrust was 36.1 mN.

A summary of all test results can be found in Table 1. Data are categorized according operating period, with the shortest at the top, and the longest at the bottom. The data in each category are arranged sequentially, with tests performed the first day being above the dashed line, and tests performed the second day below the dashed line. Second day tests are also distinguishable by higher thruster temperature, which spanned the indicated range for the longer tests. Also evident is the repeatable impulse delivered during closely spaced tests. Aside from temperature trends, most impulse values were repeatable within a few percent.

Conclusions

The 0.6 kW Electric Propulsion Demonstration Module thruster system was tested in a large vacuum facility for performance measurements and functional checkout. The thruster was operated at a xenon flow rate of 3.01 mg/s, which was supplied through a self-contained propellant system. All power was provided through a flight-packaged power processing unit, which was mounted in vacuum on a cold plate. The thruster was cycled through 34 individual startup and shutdown sequences. Operating periods ranged from 3 to 3600 seconds. The system responded promptly to each command sequence and there were no involuntary shutdowns. Direct thrust measurements indicated that steady state thrust was temperature sensitive, and varied from a high of 41.7 mN at 16°C, to a low of 34.8 mN at 110°C. Short duration thruster firings showed rapid response and good repeatability.

References

1. Lynn, P., et al.,"RHETT/EPDM Flight Hall Thruster System," IEPC Paper 97–100, August 1997.

2. Sankovic, J.M., Haag, T.W., and Manzella, D.H., "Operating Characteristics of the Russian D-55 Thruster with Anode Layer," AIAA Paper 94–3011, June 1994.

3. Sarver-Verhey, T.R., "Extended Testing of Xenon Ion Thruster Hollow Cathodes," AIAA Paper 92–3204, July 1992.

Table 01.—Summary of Results

Operating Period	Thruster Temp., C	Avg. Discharge Power, w	Average Thrust, mN	Isp, s	Delivered Impulse, mN-s	Thrust Efficiency	Time
3	22	620	40.7	1378	122	0.44	16:22
3	23	621	40.7	1378	122	0.44	16:30
3	24	613	41.0	1389	123	0.46	16:39
3	24	652	40.8	1382	122	0.42	16:47
3	89	593	39.2	1328	118	0.43	13:24
3	86	597	39.4	1334	118	0.43	13:32
10	16	633	41.7	1412	417	0.46	15:32
10	18	631	41.3	1399	413	0.45	15:41
10	18	616	41.3	1399	413	0.46	15:52
10	19	656	41.1	1392	411	0.43	16:01
10	21	626	40.8	1382	408	0.44	16:10
10	96	584	39.1	1324	391	0.43	13:06
10	93	583	39.3	1331	393	0.44	13:15
30	26	620	40.8	1382	1224	0.45	16:58
30	27	619	40.4	1368	1212	0.44	17:07
30	28	608	40.5	1372	1215	0.45	17:16
30	30	647	-	-	-	-	17:25
30	31	644	40.3	1365	1209	0.42	17:35
30	109	578	38.2	1294	1146	0.42	12:40
30	105	580	38.5	1304	1155	0.42	12:49
60	31	608	39.6	1341	2376	0.43	17:58
60	33	598	39.6	1341	2376	0.44	18:07
60	35	604	39.5	1338	2370	0.43	18:16
60	37	627	39.3	1331	2358	0.41	18:25
60	118	576	37.3	1263	2238	0.40	12:21
60	114	576	37.6	1273	2256	0.41	12:30
600	38-59	579	37.8	1280	22680	0.41	18:43
600	54-74	603	37.0	1253	22200	0.38	19:01
600	68-88	573	36.6	1239	21960	0.39	19:20
600	79-98	570	36.5	1236	21900	0.39	19:43
600	110-127	569	36.1	1223	21660	0.38	11:44
600	116-130	569	36.0	1219	21600	0.38	12:02
1200	100-129	571	35.7	1209	42840	0.37	11:07
3600	8-110	575	36.1	1223	129960	0.38	9:56

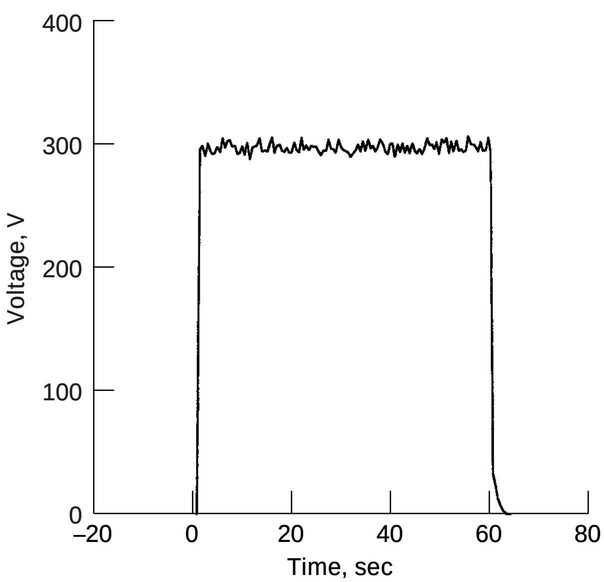

Figure 1.—Voltage as a function of time, 60 sec duration.

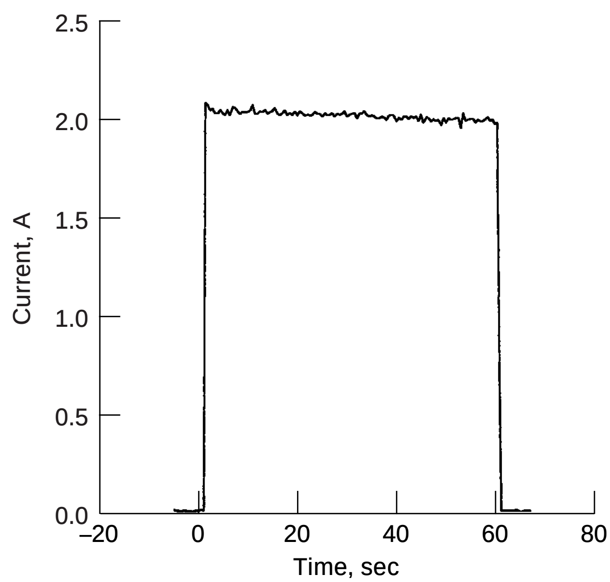

Figure 2.—Current as a function of time, 60 sec duration.

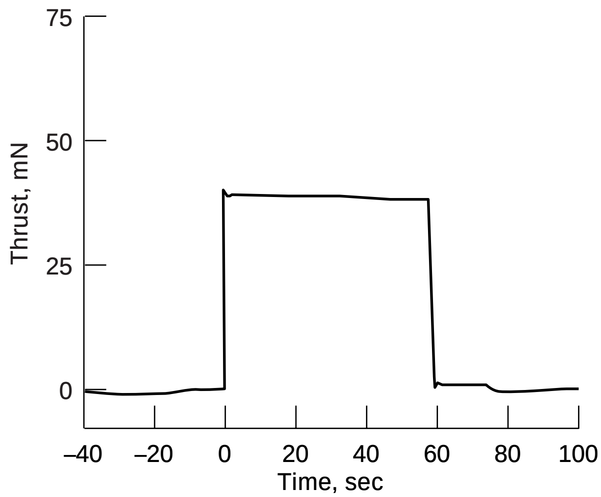

Figure 3.—Thrust as a function of time, 60 sec duration.

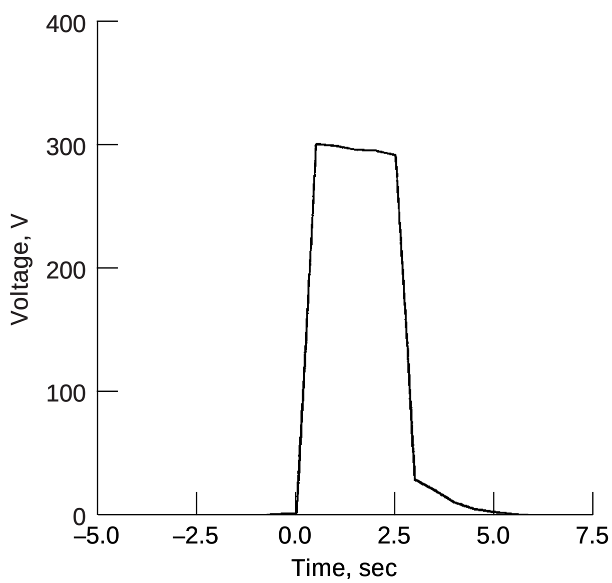

Figure 4.—Voltage as a function of time, 3 sec duration.

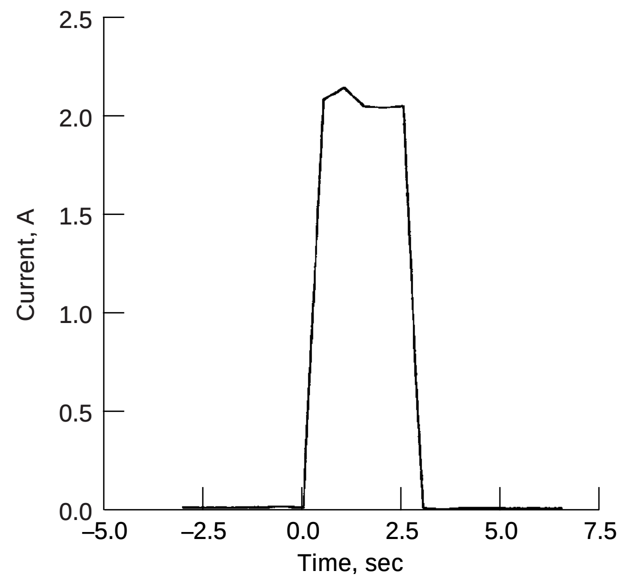

Figure 5.—Current as a function of time, 3 sec duration.

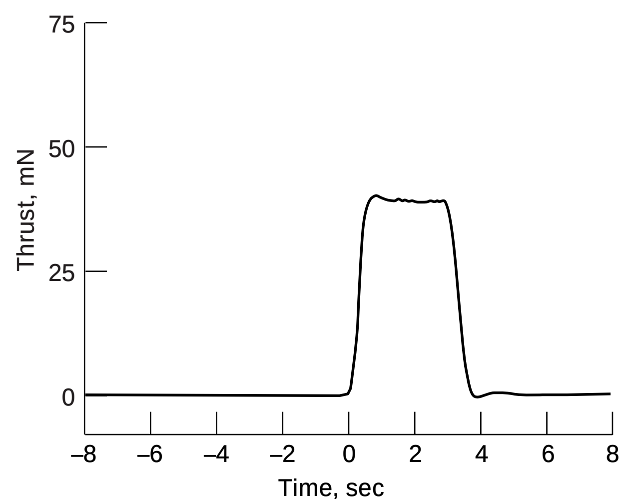

Figure 6.—Thrust as a function of time, 3 sec duration.

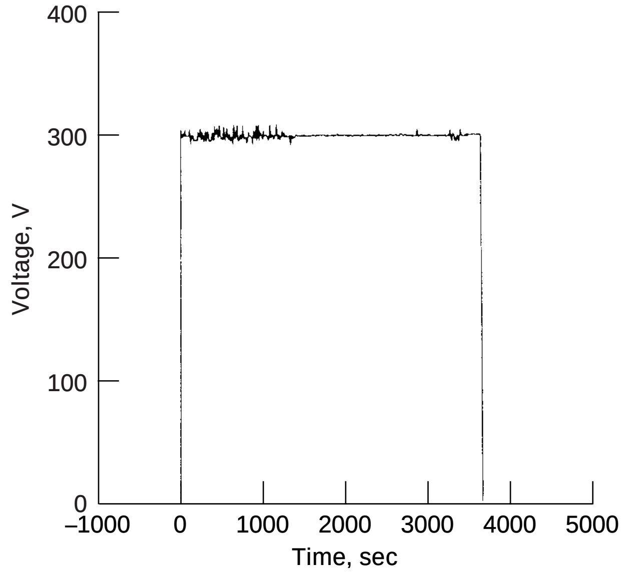

Figure 7.—Voltage as a function of time, 3600 sec duration.

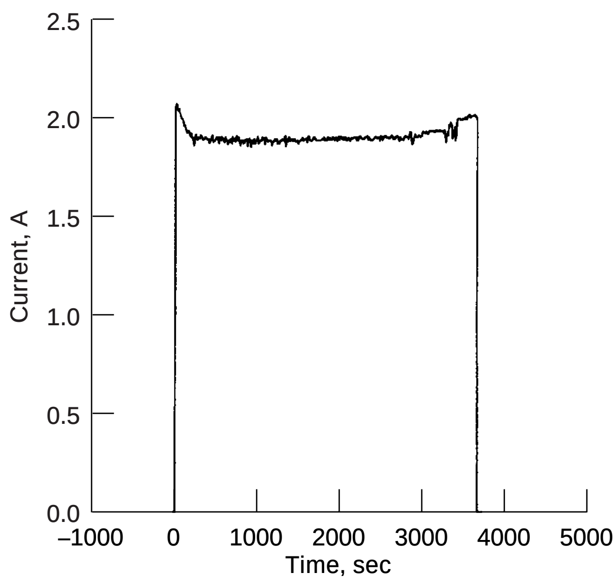

Figure 8.—Current as a function of time, 3600 sec duration.

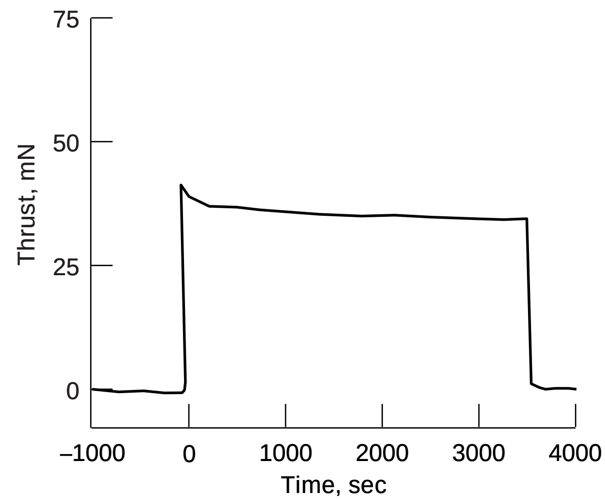

Figure 9.—Thrust as a function of time, 3600 sec duration.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)

2. REPORT DATE
June 1998

3. REPORT TYPE AND DATES COVERED
Technical Memorandum

4. TITLE AND SUBTITLE
RHETT/EPDM Performance Characterization

5. FUNDING NUMBERS
WU–632–1B–1B–00

6. AUTHOR(S)
T. Haag and M. Osborn

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135–3191

8. PERFORMING ORGANIZATION REPORT NUMBER
E–10958

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA TM—1998-206222

11. SUPPLEMENTARY NOTES
Prepared for the 25th International Electric Propulsion Conference sponsored by the Electric Rocket Propulsion Society, Cleveland, Ohio, August 24–28, 1997. T. Haag, NASA Lewis Research Center, and M. Osborn, Naval Research Laboratory, Washington, D.C. Responsible person, T. Haag, NASA Lewis Research Center, organization code 5430, (216) 977–7423.

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category: 20 Distribution: Nonstandard

This publication is available from the NASA Center for AeroSpace Information, (301) 621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

The 0.6 kW Electric Propulsion Demonstration Module flight thruster system was tested in a large vacuum facility for performance measurements and functional checkout. The thruster was operated at a xenon flow rate of 3.01 mg/s, which was supplied through a self-contained propellant system. All power was provided through a flight-packaged power processing unit, which was mounted in vacuum on a cold plate. The thruster was cycled through 34 individual startup and shutdown sequences. Operating periods ranged from 3 to 3600 seconds. The system responded promptly to each command sequence and there were no involuntary shutdowns. Direct thrust measurements indicated that steady state thrust was temperature sensitive, and varied from a high of 41.7 mN at 16°C, to a low of 34.8 mN at 110°C. Short duration thruster firings showed rapid response and good repeatability.

14. SUBJECT TERMS
Hall thruster; Space propulsion; Xenon propellant

15. NUMBER OF PAGES
15

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	